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A RESUME OF MAXIMUM LIFT DATA FOR SYMMETRICAL WINGS WITH VARIOUS HIGH LIFT AIDS

by

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SUMMARY

We shall attempt here to summarise the existing data on the values of the low-speed $C_{L_{max}}$ of wings, in the absence of a fuselage, and without including information on stalling incidence or pitching moment. The summary is limited to the consideration of unswept wings, and those of delta planform, which have symmetrical sections: there is some discussion of the maximum lift increments due to the use of flaps of various kinds.

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1. Introduction

The subject of the maximum lift of wings at low speeds of flight is one which, from time to time, occupies the attention of investigators in the field of aircraft design, because of its indirect bearing upon the cruise or top speed performance of new types of aircraft. It often happens that the take-off or landing requirement is so stringent that some sacrifice in high-speed performance is caused by its satisfaction; the sacrifice might well be alleviated by some relatively small change in wing design, but the question then arises whether quantitative results are available to show the advantages of this change.

It is with such a question in mind that we attempt here to gather together the existing data on maximum lift. The task is made much easier by the fact that it has been attempted, on perhaps a more limited scale, before; but in assembling the information from such collections of data, and from other reports of less systematic investigations, a few apparently neglected results have come to light. It is hoped that the convenience of having the information collected together, and the inclusion of these fresh results, may provide sufficient excuse for the present discussion.

So as to systemise the data, results are quoted only from wind-tunnel tests on wings alone, since it is difficult to account properly for the interference effects of a fuselage. Moreover, although the nature of the stall - as described by the stalling angle, the drag, and the pitching moment changes - is a subject of great interest and importance, no attempt is made to include data of such a kind. The prediction of stalling angle would require an analysis of the lift-curve slope of wings, which is outside the scope of the present work; and even then, notice would have to be taken of non-linear effects, which are difficult to display in any systematic form. The pitching

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moment change is likewise of great interest, but difficult to present in 'data-sheet' form.

The range of parameters covered by the analysis is generally as large as the available information permits, but as security requirements limit the amount of data which can be published on some topics - such as the maximum lift of swept wings, and boundary-layer control as a high-lift aid - such topics are in particular omitted. Further, in one or two instances, such as in an investigation of camber effects and of slotted flaps, the value of maximum lift depends critically on a large number of variables, and no complete survey of the existing data is attempted.

As a consequence of using the results of tunnel tests, certain reservations have to be made about the applicability of the data: although we try to suggest the effect of scale on maximum lift, the degree of turbulence in the tunnel is itself another source of difference to be expected in applying our results to flight in air, where the turbulence level is different, - and a measure of this discrepancy is impossible to attain. In any case, the effect of the fuselage, and other sources of interference, will modify the result in flight, where of course it may not even be possible (or desirable) to reach the stall. Nevertheless, the data given here should at least provide an adequate basis for design calculations, if interpreted realistically.

2. The Maximum Lift of Various Sections in Two-Dimensional Flow.

2.1 The Effect of Section Shape.

2.11 The Effect of Thickness.

An immense amount of data exists concerning the maximum lift of symmetrical aerofoil sections: the most successful method of reducing this data has been shown to be obtained simply by classifying each foil in terms of two geometrical characteristics: namely,

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(i) the ratio of its maximum thickness to its chord. $(=t/c)$

(ii) its nose radius, - or using a more convenient convention, the ratio of the section thickness at a station 5% of the chord aft of the nose to the maximum thickness. $(=y_5/t)$

It can be shown[1] that this system of classification collapses the data into a systematic variation, which seems to be little affected by the variation of other geometrical characteristics of section shape, and which reveals only a small experimental scatter. The result of such an analysis for symmetric sections is shown in figure 1, where $C_{L_{max}}$, the sectional maximum lift coefficient, is plotted against (y_5/t) for various ratios. The data given here includes in particular several recent test results on very thin sections.

The results will be seen to indicate the well-known result that a moderate section thickness of about 12% of the chord and a relatively thick leading edge will together produce the best value of maximum lift: an example of this type of foil is, for instance, the N.A.C.A. 0012 section* with its maximum thickness situated well forward, at a distance of 30% of the chord forward of its nose. A reduction of either the (thickness/chord) ratio, or the relative nose radius of curvature, produces a severe loss in the maximum lift which can be achieved. This is due to the different cause of stall on such sections: instead of being caused by the growth of separation of the turbulent boundary layer on the rear portion of the upper (or suction) side of the foil as on thicker sections, on thin aerofoils it is caused by the growth of a local 'bubble' of separation of the laminar boundary layer near the nose of the suction side. This latter type of stall is associated with poor lifting capabilities, due to the destruction of the region of peak suctions near the nose, and is noticed particularly on sharp nosed sections (even as thick as 10% of the chord). However, below a thickness of about 6% of the chord, all sections have a uniformly low $C_{L_{max}}$, whether or not they have a radiused nose.

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*For details of sections in the various N.A.C.A. series, see reference 3.

The change in the character of the stall occurs for most round-nosed sections if their thickness is about 9%, and it is noticeable in particular that results for these types of section display a large amount of scatter. These effects, being associated with boundary layer flow, are therefore modified to a marked degree by scale effect: the Reynolds Number appropriate to the data shown in figure 1 is 6 million.

2.12 The Effect of Camber.

A complete investigation of the maximum lift of cambered sections shows that the effect of camber cannot properly be isolated: in other words, the $C_{L_{max}}$ of a cambered section depends not only on the amount of camber and the shape of the camber line (as affecting, say, the position of maximum camber), but also on the thickness and nose radius of the section on which it is used. As a result, it is not possible to suggest, with any accuracy, the maximum lift of any arbitrarily designed, cambered, section.

The addition of camber is always beneficial to maximum lift, and the benefit grows with increasing camber. The increment to maximum lift due to camber is least for sections with relatively large radius leading edges (i.e. the benefit of camber grows with reducing y_5/t); and camber is more effective on thin sections than on thick sections. As an illustration, a 1% camber increases the $C_{L_{max}}$ of the N.A.C.A.63-021 from 1.28 to 1.31: but of the N.A.C.A.66-012 from 1.24 to 1.45. With regard to the position of the maximum camber, the evidence suggests that a forward position (up to within 15% of the chord from the nose) produces the highest values of maximum lift. For example, the N.A.C.A.23012 section (with 2% max. camber at 0.15 chord) has a higher $C_{L_{max}}$ than the N.A.C.A.4412 (with 4% camber at 0.4 chord but the same thickness distribution).

Owing to the inability to present any basic quantitative data on camber effects, the discussion

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of the following paragraphs will be limited to a discussion, in the main, of results for symmetrical wings.

2.2 The Effect of Scale.

The effect of scale upon the maximum lift is dependent on the type of stall. For symmetrical sections of moderate thickness there is a significant drop in $C_{L_{max}}$ with decreasing Reynolds' Number (below 6 million) : this is shown in figure 2 based largely on data given in references 2, 3 and 4; thin wings, on the other hand, whose stall is due to a leading-edge separation, have been found to be relatively insensitive to changes in scale, except at higher Reynolds Numbers of 20 million or so where $C_{L_{max}}$ appears to increase.

The data shown in figure 2 are based to a large extent on low-scale pre-war tests on 'conventional' type sections, whose geometry involves a varying (t/c), but a uniform value of about 0.6 for (y_5/t); the data are not sufficient to isolate the effect of variations in nose thickness on the scale effect, but one forms the impression that probably the effect is much more dependent on the value of (y_5/c) than on (t/c), (i.e. on nose radius rather than overall thickness) and it is suggested that the data given be interpreted with this in mind.

The evidence points to the fact that at low Reynolds Numbers (below about a million) the maximum lift of all sections of whatever thickness is uniformly low (due to leading edge separation).

Scale effects are less for cambered wings than on symmetrical sections so that at low Reynolds Numbers, the benefit of camber is much more marked: the opposite is true at Reynolds Numbers greatly in excess of 6 million, where camber loses some of its effect.

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2.3 The Effect of Surface Condition.

Experiments on model aerofoils which have been deliberately roughened above the tolerable limit (either over their whole surface, or just over the leading-edge region) have shown a significant loss in maximum lift: this is exemplified in figure 3, the data for which have been obtained mostly from reference 3. It will be seen that losses of up to 0.5 in $C_{L_{max}}$ are suffered by sections of moderate thickness (which in the smooth condition produced a very high $C_{L_{max}}$). This is presumably because the presence of roughness has a destabilising effect on the turbulent boundary layer, causing a premature break-away of the flow.

On the other hand, thin sections are hardly affected at all by roughness - in some cases the $C_{L_{max}}$ is even improved: this is presumably because the local turbulence produced by the roughness has a stabilising effect on the region of 'bubble' separation, where the boundary-layer flow would otherwise be laminar (and therefore less stable). The same effect is also noticed on all sections at low enough Reynolds Number (when laminar breakaway is present), and for this reason (as exemplified in figure 4) the scale effect on rough sections is much less severe than in the smooth condition, - with rough sections in many cases having a little higher value of sectional maximum lift than the identical smooth sections at low Reynolds Numbers.

Thus we remark that, if the working Reynolds Number is of the order of several million, then there is no particular merit in choosing a 'high-lift section' unless we can ensure that the leading-edge is smooth. On the other hand, if for other reasons the section chosen is a thin one, or if the scale is low, there is no particular merit in ensuring that the surface is smooth - at least as far as the maximum lift is concerned.

The results on which figure 3 are based are derived from N.A.C.A. tests on aerofoils with a 'standard roughness' : although it is true that the rougher the section, the more marked will be the effects mentioned, the results for 'standard roughness' represent about the greatest effect found in practice. Little or no information exists on the effects of waviness of the surface on $C_{L_{max}}$, although it is well known that small errors in the geometry of models have led to large losses in maximum lift - and in fact, in some cases, these errors have been so small that they have only been detected because of the loss observed.

2.4 The Effect of Trailing-edge Thickness.

A parameter which was not considered in our discussion of the effects of the geometry of various aerofoil sections is the thickness of the trailing-edge. However, lately the introduction of a blunt, cut-off, trailing-edge has been found to lead in certain cases to the reduction of the wave drag at supersonic speeds, and consequently the low speed characteristics of such sections have been studied; it has been found that the thickening of the trailing-edge can have a beneficial effect on $C_{L_{max}}$ (see, for example, reference 5). A disadvantage of this thickening lies in the consequent presence of buffetting of the flow behind the wing : however in certain circumstances, this might be no embarrassment.

In figure 5, a curve is given indicating the effect of thickening the trailing-edge. The available data are hardly sufficient to suggest that the effect will be the same on all sections, and for that reason, the data of figure 5 must be regarded as purely tentative. However, it seems to indicate that a significant increase in $C_{L_{max}}$ may be effected by thickening a wing section aft of its maximum thickness, and making the trailing-edge thickness finite. In this process it is assumed that the parameters (t/c) and (y_5/t) of the section are unaltered : and the figure shows the increment in $C_{L_{max}}$ obtained.

It does not follow that a section formed by cutting-off a basic section, (at some chordwise position in front of the trailing edge of the basic section) would necessarily have a higher $C_{L_{max}}$. This is illustrated by figure 6 for the N.A.C.A. 0012 aerofoil section. If the section shape were changed, by thickening aft of the maximum thickness position of the basic section (at 0.30 c), so that a uniform thickness were maintained from the 30% chord point to the trailing edge, we would expect on the basis of figure 5 that the $C_{L_{max}}$ might be increased from 1.60 to 1.97, say. However if the N.A.C.A. 0012 is cut off at its maximum thickness position, although this new section has a trailing-edge thickness equal to its maximum thickness, it is in other respects an entirely different aerofoil : it has a thickness/chord ratio of 40%, and the value of (y_5/t) is changed from 0.59 to 0.35 ; its maximum lift will presumably be very low. Thickening a section is one thing - and in general a beneficial modification: cutting-off a section is another thing altogether.

As far as the limited data permit us to draw any conclusions, it appears that the addition of a trailing-edge thickness is just as effective on rough surfaced wings as on smooth.

3. The Effect of Planform on Maximum Lift.

This paragraph deals chiefly with the performance of wings of very low aspect ratio, and it is realised that their capabilities are not so much governed by the maximum lift attainable - which is the property we are discussing - but by such considerations as what incidence may reasonably be achieved, and what degree of trim change can be accomodated. For the high-lift investigations of such wings show that the maximum lift is reached at very high incidences(30° and more), and that at some stage long before the stall, there is a marked rearward shift in aerodynamic centre, ultimately contributing to a large nose down change of trim. Nevertheless the values of the true maximum lift show astonishing differences from the sectional values, and

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a consideration of these is of some interest to the designer (if only in relation to tailplane design).

3.1 Unswept Wings.

By an unswept wing we mean here one whose half-chord line is (at least roughly) perpendicular to the direction of motion.

Speaking first of wings of moderate thickness, it is known that reduction of aspect ratio (down to about 2, say) results in a marked drop in $C_{L_{max}}$ compared with the two-dimensional value (or sectional maximum lift, corresponding to an aspect ratio of infinity). This drop due to aspect ratio may be accounted for, at least qualitatively, by supposing that the stall is 'precipitated by the rapid growth of a local separation in that region of the wing where the local value of C_L reaches the two dimensional maximum'. With reduction in aspect ratio, the loading at the centre line becomes relatively higher (with a local C_L in excess of the wing total or mean, lift coefficient) so that the stall occurs at a lower mean C_L. This description is fairly adequate, except for wings of low aspect ratios - below, say, about 3, - and except for wings with very thin sections : these exceptions, of course, would often include wing designs envisaged for use on supersonic aircraft or missiles.

We consider first the case of the thin wing, represented in figure 7 by results for flat plates taken from References 6-8. The effect of aspect ratio is displayed by subtracting the measured $C_{L_{max}}$ from that for the same 'flat plate' of infinite aspect ratio (which in most cases is around 0.7).

Down to an aspect ratio below 2 there is a well marked increase of $C_{L_{max}}$ with aspect ratio. This reaches a maximum, for most planforms, where the span is roughly equal to the root chord. For wings of elliptic planform for instance, the highest $C_{L_{max}}$ is obtained on a circular wing. At smaller aspect ratios

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the rise is arrested, and it is observed that (for most of the incidence range) the flow over the centre of the suction-side is separated, with the wing tip vortices forming a strongly developed 'cushion' on either side of this region. At still lower aspect ratios, the stalling angle is about 45^{o} and the maximum lift drops (as a result of the decreasing lift curve slope with reduced aspect ratio) ; here the stall is very gradual, and is marked by the fact that the tip vortices, instead of forming a 'cushion' on the wing, break away from the upper surface at positions progressively nearer the leading-edge, as the incidence is increased. It seems that the initial increase in $C_{L_{max}}$ with decreasing aspect ratio is caused by the importance of the tip-flow : the tip vortices grow rapidly downstream and their induced downwash which is greatest near the trailing edge, reduces the likelihood of complete boundary layer breakaway.

On thick wings, the development of the tip-vortex system, with the ultimate confined region of separation near the centre on the suction side, is much delayed. As will be seen from figure 8, assembled from the data of references 7-11, the rise in $\Delta C_{L_{max}}$ is delayed until the aspect ratio is near 1.5, and the extent of the rise is consequently less. Here in figure 8, $\Delta C_{L_{max}}$ is obtained by subtracting from the measured value of $C_{L_{max}}$ that corresponding to the two-dimensional value for the same section at the same scale* ; no methodic change with

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*The sections used were all about 12% thick, with and without camber : the test Reynolds Numbers ranged from 0.3 to 1.6 million. In the interpretation of the data one important point should be noted : for aspect ratios around unity, there are often two maxima in the curve of C_L versus incidence : the first 'stall' (occurring at an incidence of about 20^{o}) is soon followed by a recovery of lift, but not all investigators pursued their experiments so far as to notice this : to make the data more realistic, the figure for the higher stall has been selected if this is observable. Again (at aspect ratios around $\frac{1}{2}$) often a first 'stall' occurs at about 40^{o} only to be followed by a sudden rise in lift well above its previous maximum, with a second stall at around 50^{o} : in this case, however, the second stall is neglected, admittedly merely because its appearance is so unsystematic, and its effect so inordinate.

aspect ratio in scale or camber effect on the value of $C_{L_{max}}$ is evident from the limited data, but the scatter is large and the presence of such changes cannot be ruled out. However, there appears, as for the flat-plates, to be a significant difference in the results for elliptic planforms : the greatest value of $C_{L_{max}}$ occurs again for a circular wing (and makes one wonder whether the 'flying saucer' may not have its advantages!).

It is difficult to be precise on the basis of the data given in figure 8, but if we compare these results with those of figure 7 - noting that for most of the thick sections tested, the two-dimensional value of the maximum lift coefficient is around 1.6 at a Reynolds Number of 6 million - it will be seen that at low aspect ratios the maximum lift of the thick wings is not greatly in excess of that for flat plate sections. Indeed, at a smaller scale, the thicker 'high-lift' wings are appreciably worse than the flat-plates. This seems well established, at least qualitatively, by the data : the reason for it is obscure.

3.2 Wings of Delta Plan Form.

The available data on wings with delta planforms - i.e. all those with straight trailing-edges - have been collected and studied. As yet, the requirements of security leave the picture incomplete, and the only data shown here are those for such wings of flat plate section[7,8] (figure 9). The increment in $C_{L_{max}}$ is again plotted as a function of aspect ratio, and the planforms tested fall into two distinct categories : the true delta (or triangular) planforms, and 'cropped' deltas (some having a finite tip chord, and others having curved, elliptical, leading-edges). Together with the previous results for rectangular sections, the data show in fact the effect of tapering the wing, leaving the trailing edge unswept. At high aspect ratios (above about 2) the effect of such taper is to improve the value of $C_{L_{max}}$, but at smaller aspect ratios (below unity) the reverse is true, as will be seen from figure 9.

Not enough data is available to draw similar curves

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for wings of moderate thickness and delta planform, but the same relative tendencies due to taper are observable as for the flat plate (see, for example, reference 19) : initially there is drop in $C_{L_{max}}$ due to decreased aspect ratio - though not so severe as that on a rectangular wing of moderate thickness - followed by a rise (at aspect ratios around 2), and then a second gradual fall, showing the triangular wing of moderate thickness to have a lower $C_{L_{max}}$ than the rectangular one at low aspect ratios (below unity). Cropped deltas of moderate thickness trace an intermediate path between the results for triangular and rectangular planforms.

3.3 The Effects of Sweepback.

It is impossible within the confines of this note to discuss in any detail the general effects of sweep on $C_{L_{max}}$, - including all the results on wings with both leading and trailing edges sweptback or forward to any appreciable extent. Although the results are numerous, they are not yet fully understood, and certainly there is a significant variation due to scale effect. As a general rule, it is observed that a small amount of sweep has a beneficial effect on $C_{L_{max}}$, whereas a large sweep can be decidedly detrimental ; this bears out the results of figure 9, since triangular wings of high aspect ratio are less swept-back than those of low aspect ratio, and the former have a better - whilst the latter have a worse - value of maximum lift than the unswept fully tapered planforms of equivalent aspect ratio. However the effect is much more complicated than such a simple rule would suggest, and in any appreciation of the aerodynamic data it must be noted that many types of highly swept planforms have large values of $C_{L_{max}}$, but have been comparatively neglected by investigators, because of undesirable stability characteristics.

3.4 The Effect of Endplates.

We might expect the addition of end plates to a wing to have some small effect on $C_{L_{max}}$, since they will change the effective wing aspect ratio. If the end plate is sufficiently large the change in $C_{L_{max}}$ observed as due to their inclusion can be satisfactorily accounted for on

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this basis : however, there is usually an unfavourable interaction between the boundary layer flow over the end-plate and that over the wing-tip which, particularly if the end plate is relatively small, can cause an unexpected loss in $C_{L_{max}}$ of larger magnitude than the gain (or loss) due to the change in effective aspect ratio.

4. High Lift Aids.

4.1 The Plain Flap.

This is the simplest type of trailing-edge device and is formed merely by hingeing the trailing edge portion of the wing, as shown in figure 10(a). Generally, the sectional (or two-dimensional) maximum lift coefficient increases with deflection of this flap up to about 60° or 70°, except where the chord of the flap exceeds 0.3 of the wing chord (when the angle is less).

The data are rather meagre (and derived chiefly from reference 12) but show that the best sectional maximum lift coefficient is attained with flaps of 0.2 or 0.25 of the wing chord ; how rapidly it deteriorates for high chords is not well established.

With the optimum deflection and flap chord, the value of the sectional maximum lift coefficient depends essentially upon the geometry of the section, and is most sensitive to the value of the leading edge radius, which we can conveniently describe by the parameter (y_5/c), as shown in figure 11. The value of $C_{L_{max}}$ with flap deflected is seen to increase with increasing nose radius up to a maximum at $(y_5/c) = 0.10$ approximately : thereafter the increment in sectional $C_{L_{max}}$ due to flap appears to remain at about 1.0, and the total $C_{L_{max}}$ drops.

As far as the data permit any conclusion to be formed, it is that the increment in maximum lift coefficient due to the flap deflection is insensitive to changes in scale or surface condition (although of course the total maximum lift coefficient will be changed in the manner previously described for sections without flaps). There is little

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data, too, on the effect of change in maximum lift increment with aspect ratio : at high aspect ratios (and with a wing section of moderate thickness) the increment decreases with reducing aspect ratio, roughly in the same proportion as does the maximum lift of the wing with the flap undeflected; but there is no information relevant to aspect ratios less than about 3. Whether it is also true that the increment increases with reducing aspect ratio on a thin wing - as does the maximum lift of the wing with flap undeflected - is not known.

One important point must be borne in mind in the design of plain flaps : a gap between the airfoil and the flap at the flap hinge allows air to leak through from the high pressure on the lower surface to the low pressure region on the upper surface, with a consequent severe loss in flap effectiveness. Even with so minute a gap as (1/300)th of the chord a loss of 0.35 has been observed in $C_{L_{max}}$.

4.2 The Split Flap.

At chords and deflections corresponding to the optimum values for a plain flap, the split flap (figure 10(b)) will produce a slightly larger increment in sectional $C_{L_{max}}$. (see figure 11 from reference 1) This is due to the fact that the upper surface contour is undisturbed by deflection of a split flap, and the region of separated flow at the hinge of the plain flap, where there is an abrupt downward curvature of the surface, is consequently avoided.

If this were the only advantage of the split flap (in terms of increment to $C_{L_{max}}$) it would be hardly worth the structural complication. However, because the flow over the flapped part of the section has a tendency to remain unstalled up to higher deflections, by increasing both the flap deflection and its chord appreciably higher increments may be reached. This is shown by the results of figure 12 (taken chiefly from reference 12), where it will be seen that generally the maximum lift is increased by extending the flap chord to some 40% of the wing total chord, and by operating at optimum deflections which are higher than is appropriate for a plain flap. The split flap appears

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to function most efficiently on a wing section with a fairly high leading edge thickness (with $y_5/c = 0.10$), but the data are somewhat limited for sections with thicker noses.

As will be seen from figure 13 some reasonably methodic data exist for the scale effect on the maximum lift of split flaps. In this figure (assembled mostly from data in references 4 and 12) some results are plotted showing the <u>change</u> in the sectional maximum lift <u>increment</u> due to the flaps : these changes, in other words, are additional to those for the effect of scale on wings without flaps (given in figure 2). These data appear to be dependent on the (thickness/chord) ratio of the section, in as much that the 18% thick sections show a significant loss in flap effectiveness below a Reynolds Number of about 3 million, whereas progressive reduction in thickness of the section leads to an increase in flap effectiveness with reduced scale.

These scale effects refer to wings in a smooth condition: the maximum lift of a wing with a split flap also appears to be adversely affected by roughness and some results are shown in figure 11, (from the data of references 3 and 4). The effects of scale on rough sections are more difficult to analyse : most flaps lose a little of their effectiveness at low Reynolds Numbers*.

As for plain flaps, the data on aspect ratio effect (at least in the lower range) is virtually non-existent, and it is difficult even to guess what it might be.

Some data exist for extensible split flaps (whose hinge line moves aft when the flap is deflected). These derive some extra benefit due to the increased section

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*At low Reynolds Numbers one would anticipate little difference in overall performance whether the wing were rough or smooth. This is an entirely general conclusion, but it does imply that the loss in flap effectiveness in the rough condition is less - with change in Reynolds Number - than for the smooth wing.

chord : and their quantitative effect can be estimated with sufficient accuracy on this basis.

4.3 The Slotted Flap and External Aerofoil.

The most efficient type of trailing-edge device is undoubtedly the slotted flap, (figure 10(c)). There is little point here in attempting to predict the performance of a slotted flap as so much depends on the detailed design of the slot entry and lip : at best, the maximum lift performance is a little better than the split flap, and certainly the drag is less ; at worst it is merely an inefficient plain flap, with the slot acting as an unwanted gap.

A neglected, but rather simpler, form of slotted flap is the external aerofoil type shown in figure 10(d), which has the added advantage over the split and slotted flaps that it can also be used as a control. This flap consists of a small aerofoil mounted in tandem with, and immediately behind, the main section. On deflection of this external foil, a gap is opened between the trailing edge of the main section and the leading edge of the flap: the presence of this gap (if of the appropriate size) permits the boundary layer on the lower surface to blow through and re-energise the boundary layer on the top surface of the flap, which would otherwise separate. The gap, in other words, fulfils the function of the slot, though not so effectively as a properly designed slot. As a control it performs as well as the usual type (without the gap) but suffers from the disadvanatge that special precautions would be needed to overcome icing troubles.

No methodic data are available for the external aerofoil flap, but figure 14 shows some results in a typical experiment[13]. The increment, due to flap deflection, in the sectional maximum lift will be seen to be better than could be attained from a plain flap, but a little less than from a split flap. The sensitivity to gap size of the increment in $C_{L_{max}}$ is clearly demonstrated, and in view of the lack of data, detailed tests on each new design of wing would have to be made (in the first cases) to

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establish the optimum size. Its deflection is also important : as distinct from the behaviour of split and plain flaps, exceeding the optimum angle of deflection of an external aerofoil flap produces a severe breakdown in lift and a rise in drag. The great aerodynamic advantage of this flap is its relatively low drag, when deflected, compared with the other trailing-edge devices.

4.4 The Nose flap.

The leading-edge plain flap (shown in figure 10(e)) is not so effective as the trailing-edge plain flap, but may be used in conjunction with the latter to increase its effect. Nose flaps are most effective on thin sharp-nosed sections where the deflection of the nose reduces the high suctions which otherwise exist in this region : some representative results for the increment in sectional maximum lift are given in figure 15 from reference 14. The increments are somewhat lower when the flap is used in conjunction with a trailing-edge flap, but data on the combination are less complete.

The leading-edge flap has not been examined in any detail for thicker sections : an isolated result for a 12% thick, bluff-nosed section[15] shows an increment in C_{Lmax} of just over 0.2 for a 15% chord flap (as compared with over twice this figure for the thin wing shown in figure 15). Nor is there any information on scale or aspect ratio effects.

There are many other types of nose flap, such as the Kruger flap shown in figure 10(f), which derive some extra benefit either by increasing the wing chord or by an effective increase in the nose radius of curvature. For thin sections the first cause explains their quantitative advantage over the simple hinged type : however for thicker sections (with t/c greater than, say, 6%) a further advantage lies in the increase of nose radius, which as we see from figure 1 plays an important role in determining maximum lift. This latter effect is of course particularly obvious for relatively sharp nosed sections with thickness chord ratios of from 6 - 12% of the chord.

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4.5 The Leading Edge Slot, and Front Auxiliary.

The leading edge slot (figure 10(g)) has long been known as a means of improving the maximum lift of a section, merely by postponing the stalling angle. The analysis of reference 16 suggests that a slot set at about 40° to the wing chord line will delay the stall to an extent of about 10° and provide an increment to maximum lift of about :

$$\Delta C_{L_{max}} = 3.3 \text{ (slot chord/wing chord)}$$

provided the slot chord does not exceed about 30% of the main wing chord. These increments, moreover, are additive to those obtained from the trailing-edge flaps, if fitted.

A slotted leading edge flap - that is, one which (like the trailing-edge slotted flap) opens a gap on its deflection - has apparently no great advantages over the simple leading edge plain flap, as is suggested by the following table (from the data of reference 17) :

Type of flaps used (0.164c leading-edge flap, and 0.254c trailing-edge flap)	Value of $C_{L_{max}}$			
	With no flaps	With leading edge flap only	With trailing edge flap only	With both flaps
Plain	0.65	1.04	1.39	1.84
Slotted	0.65	-	1.55	2.02

The increment in $C_{L_{max}}$ due to the addition of leading-edge flaps on the wing with trailing-edge flaps is nearly the same, whether the former are plain or slotted. The wing to which these results refer was one of aspect ratio $5\frac{1}{3}$ with a $7\frac{1}{2}\%$ thick biconvex section. No other data on slotted leading-edge flaps has been found.

A different type of leading edge slotted flap is the front auxiliary aerofoil, shown in figure 10(h) : like the external aerofoil flap, it is a small aerofoil mounted in tandem with, but forward of, the main section. Also like the rear external aerofoil, it has received little attention, and only some particular results[18] may be quoted (figure 16) : however these are of particular interest.

/Unlike ...

Unlike the rear auxiliary, it is comparatively insensitive to the gap size, which will be seen to be best made quite large : moreover the optimum angle of deflection was found to be always within a few degrees of zero for all the medium sized auxiliaries tested in reference 18. Increments of about 0.5 in the sectional maximum lift coefficient are obtainable from all the auxiliaries, of whatever size ; but whether this would be additive to that obtained from a trailing-edge flap, used in conjunction with tho auxiliary, is a point not yet investigated.

In view of the comparative insensitivity of the device to changes in the many design parameters involved, there might be nearly as much benefit, - without so much structural complication, - in fixing the auxiliary in a position of symmetry ahead of the main section : some idea of the $C_{L_{max}}$ increments obtained in this way is indicated in figure 17 from the data of reference 18, (where admittedly the main section used was cambered and therefore a true position of symmetry does not exist).

The front auxiliary derives its effect by inducing a downwash which is largest near the leading-edge of the main section, reducing the local intensity of suction and delaying boundary layer separation (including, presumably, on thin wings, the bubble form). On the other hand, the main section induces a large upwash over the auxiliary, which does not however stall so readily as the main wing because of the beneficial 'blowing effect' of the flow through the gap on its boundary layer : consequently very high normal forces are experienced on the auxiliary near the stall of the combination (normal force coefficients as high as 3 having been recorded on this auxiliary, without any sign of a loss in effectiveness!).

No data again is available on thickness, scale or aspect ratio effects : like most leading-edge

/devices ...

devices, we would expect most benefit to accrue from its use on thin sections, and least on thick sections. But of all the devices which for particular reasons have been neglected in aeronautical development, this would appear the simplest and most worthwhile of investigation as a high-lift aid, particularly for high speed aircraft, whose thin wings and symmetrical sections make the provision of leading-edge slots otherwise difficult.

The reasons for this neglect appear to be relevant only to the context in which it was originally conceived by the N.A.C.A. - as a substitute for aileron controls, as well as a high lift aid; - as a control it introduced a number of stability troubles, and although tested in flight, it was abandoned at an early stage in favour of trailing-edge controls. However merely as a fixed high-lift aid, it has the advantages of structural simplicity, and its properties seem to be worthy of further investigation. Its use on supersonic or transonic aircraft suggests no disadvantage to the performance, but buffetting caused by the wake of the auxiliary on the main wing might be present, and if so this would be a severe embarrassment.

Conclusions.

1) The effect on $C_{L_{max}}$ of the thickness distribution may be displayed by a consideration of nose radius and maximum thickness, as shown in figure 1.

2) The effects of camber on $C_{L_{max}}$ are difficult to isolate, but always lead to an increase of maximum lift particularly on the thinner sections, and those with relatively small nose radii. A forward position of maximum camber is advantageous.

3) Scale effects for symmetrical wings are shown in figure 2, but most of the low Reynolds Number data exists for the conventional types of wing section (with the maximum thickness well forward). However nose thickness is probably more important than maximum thickness.

/ 4) ...

4) The presence of roughness has a detrimental effect on 'high-lift' wing sections, but can be beneficial on thin wings (figure 3 - 4).

5) The thickening of the wing section aft of the maximum thickness, so as to leave the trailing-edge blunt, improves the $C_{L_{max}}$ (figure 5).

6) Reduction of aspect ratio generally improves the $C_{L_{max}}$ of thin sharp-nosed unswept wings, at least down to aspect ratios of unity. A circular planform appears to have a particularly high maximum lift (figure 7).

7) Reduction of aspect ratio of a moderately thick unswept wing (with t/c around 12%) progressively reduces $C_{L_{max}}$ until an aspect ratio of about $1\frac{1}{2}$ is reached. Further reduction produces a sudden recovery in maximum lift, followed by a gradual fall. At very low aspect ratios, thickness appears to have lost much of the effectiveness it has at high aspect ratios in governing the value of $C_{L_{max}}$ (figure 8).

8) A delta wing has a higher $C_{L_{max}}$ than an unswept wing of the same aspect ratio, at least down to an aspect ratio of about $1\frac{1}{2}$: thereafter it has a rather lower value (figure 9).

9) There is a lack of data on the effect of aspect ratio on the maximum lift increments due to flaps, which presents an obstacle to satisfactory prediction of $C_{L_{max}}$ characteristics of aircraft.

10) Both plain and split flaps are most effective on wing-sections whose thickness, 5% of the chord aft of the nose, is about 0.1 of the chord.

11) Fairly comprehensive data exist for the performance of split flaps (figure 12) and some reasonably methodic data on scale effects have been found (figure 13).

12) Some data are presented for the external aerofoil trailing-edge flap, in figure 14, but it is a device which is particularly sensitive to detailed design.

/ 13) ...

13) The front auxiliary aerofoil appears to be a neglected, but particularly simple and effective, high-lift aid (when either hinged or fixed) which might have useful applications (figures 16 and 17).

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/ 10. ...

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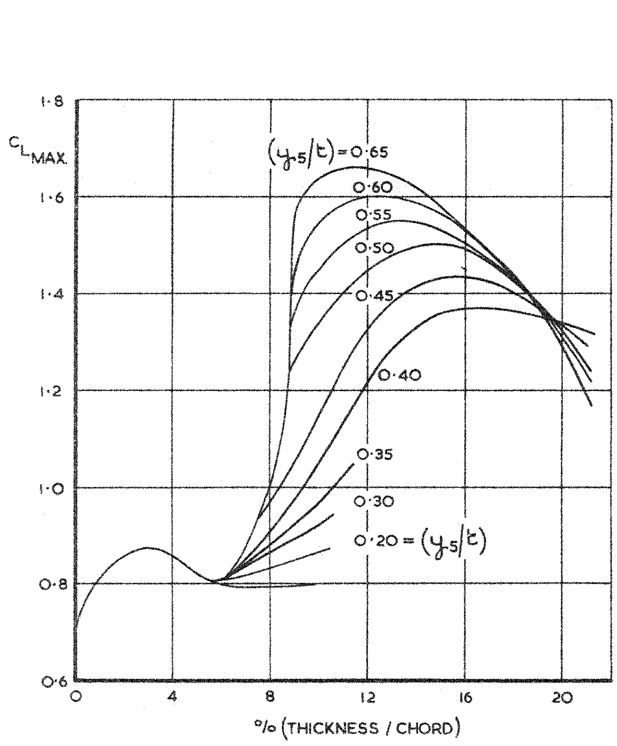


THE VARIATION OF SECTIONAL MAXIMUM LIFT WITH THE GEOMETRY OF SYMMETRICAL AEROFOILS

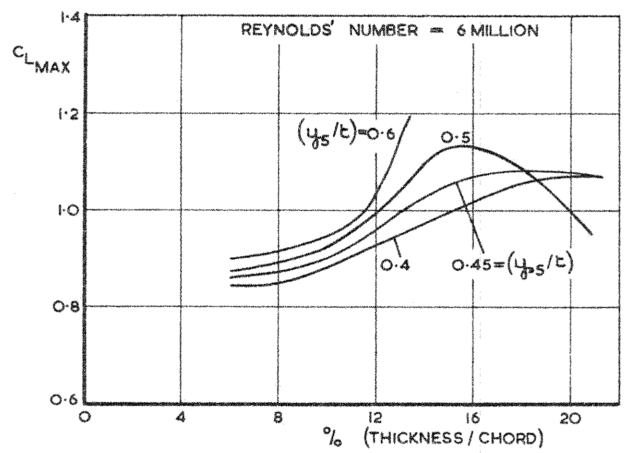
FIG. 1.

y_5 = THICKNESS OF SECTION 5% OF THE CHORD AFT OF NOSE.

t = MAXIMUM THICKNESS OF SECTION.

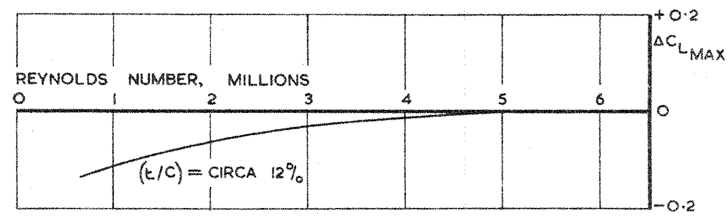
REYNOLDS NUMBER 6 MILLION.

SURFACE CONDITION — SMOOTH.



THE MAXIMUM LIFT OF ROUGH SURFACED SYMMETRICAL AEROFOILS

FIG. 3.



SCALE EFFECT ON ROUGH SURFACED AEROFOILS

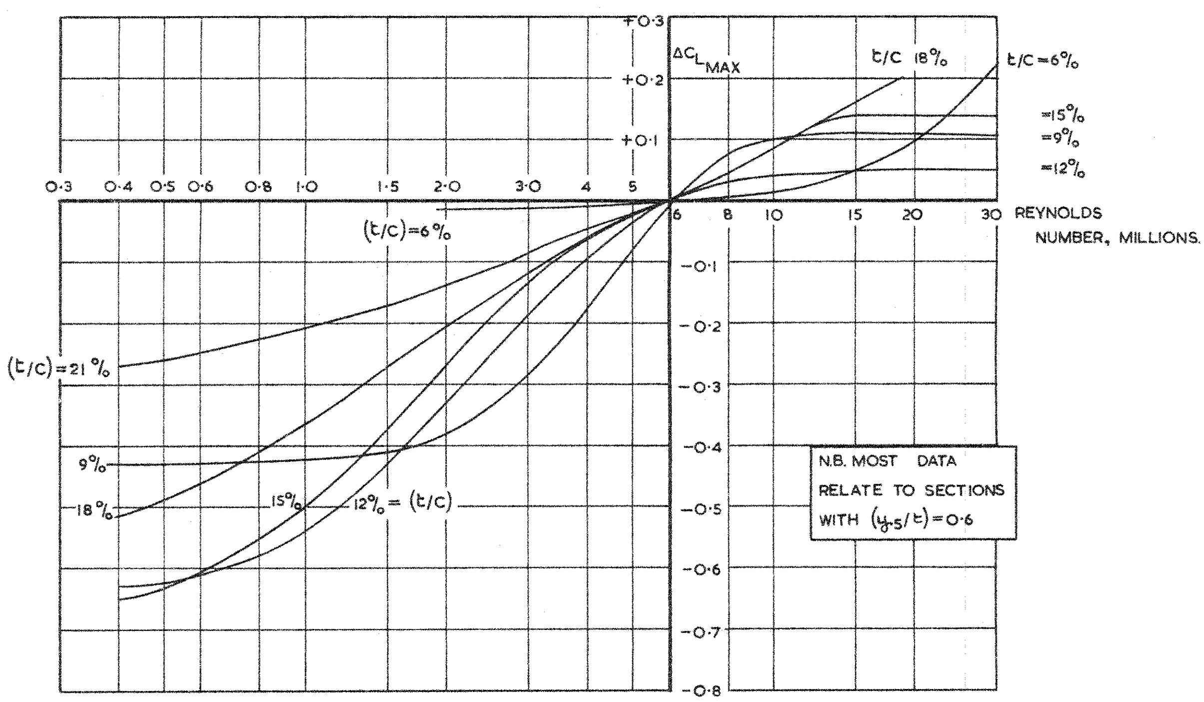
FIG. 4.

y_5 = THICKNESS OF SECTION 5% OF CHORD AFT OF NOSE.

t = MAXIMUM THICKNESS OF SECTION.

C = CHORD OF SECTION.

$\Delta C_{L_{MAX}}$ = CHANGE IN $C_{L_{MAX}}$ RELATIVE TO DATUM AT REYNOLDS NUMBER OF 6 MILLION.



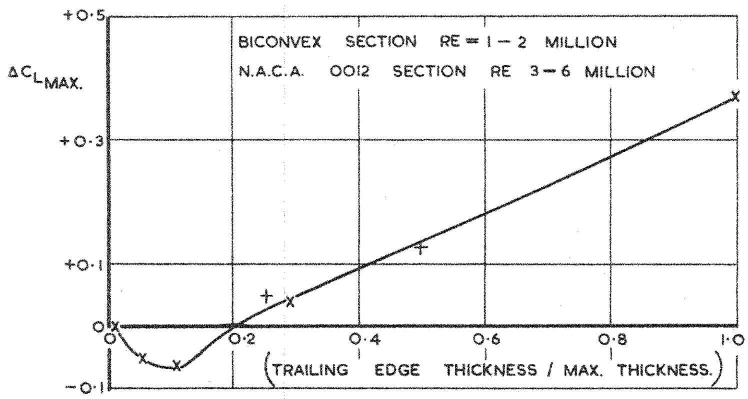
THE EFFECT OF REYNOLDS NUMBER ON SECTIONAL MAXIMUM LIFT

SURFACE CONDITION — SMOOTH

FIG. 2.

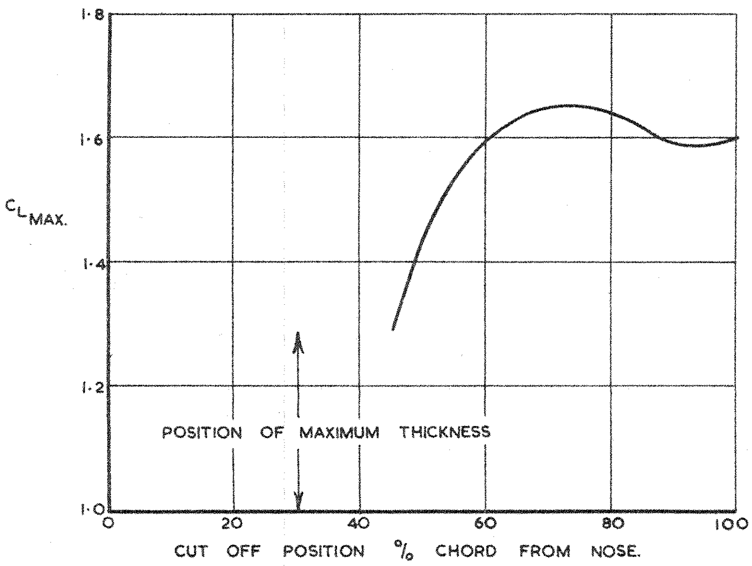
t/c = THICKNESS/CHORD RATIO.

$\Delta C_{L_{MAX.}}$ = CHANGE IN $C_{L_{MAX.}}$ RELATIVE TO DATUM AT REYNOLDS NUMBER OF 6 MILLION



INCREMENT IN $C_{L_{MAX}}$ CAUSED BY THICKENING A SECTION AFT OF ITS MAXIMUM THICKNESS.

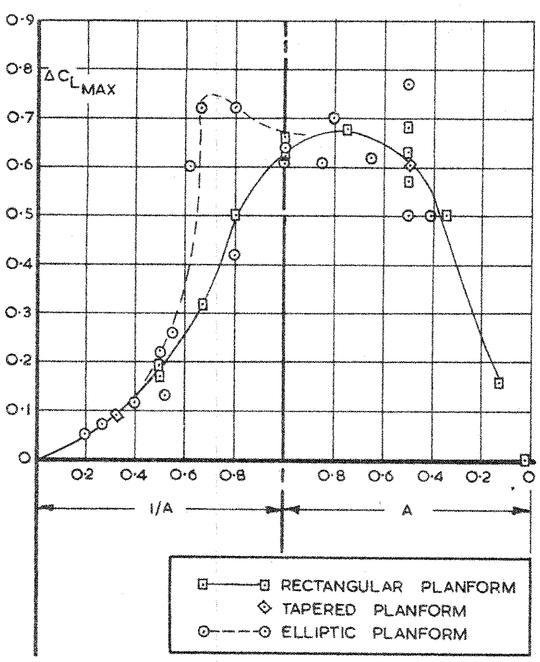
FIG. 5.



INFERRED SECTIONAL $C_{L_{MAX}}$ FOR A SMOOTH N.A.C.A. 0012 SECTION WITH VARIOUS POSITIONS OF THE CUT-OFF.

REYNOLDS NUMBER 6 MILLION

FIG. 6.

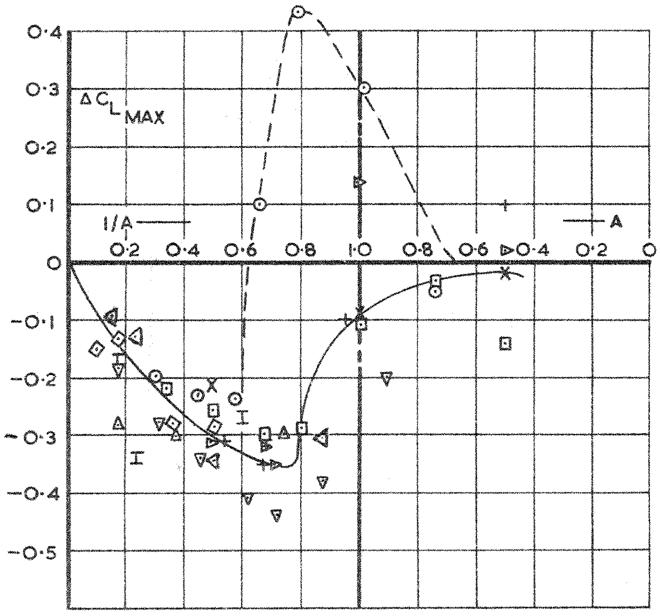


THE EFFECT OF ASPECT RATIO IN THE MAXIMUM LIFT OF UNSWEPT WINGS OF FLAT PLATE SECTION.

($\Delta C_{L_{MAX}}$ = INCREMENTAL CHANGE IN $C_{L_{MAX}}$ RELATIVE TO SECTIONAL VALUE AS DATUM.

A = EFFECTIVE WING ASPECT RATIO.)

FIG. 7.



THE EFFECT OF ASPECT RATIO ON THE MAXIMUM LIFT OF UNSWEPT WINGS OF MODERATE THICKNESS

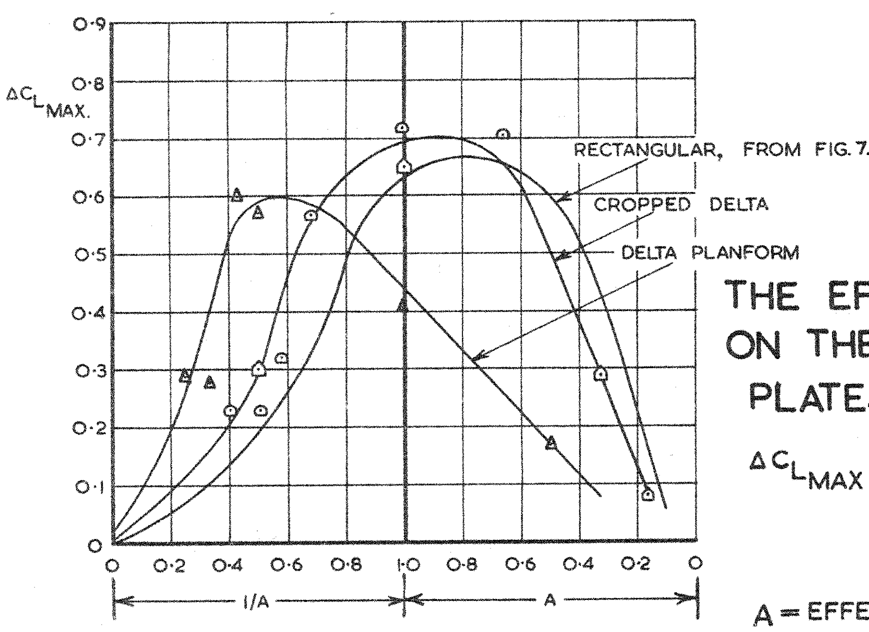
$\Delta C_{L_{MAX}}$ = INCREMENTAL CHANGE IN $C_{L_{MAX}}$ RELATIVE TO SECTIONAL VALUE AS DATUM.

A = EFFECTIVE WING ASPECT RATIO.

FIG. 8.

KEY TO DATA.

PLANFORM	WING — SECTION.				
	CLARK Y	N.A.C.A. 0012	R.S.G. 35	GOTT. 409	N.A.C.A. 64010
RECTANGULAR	⊡	X	I	+	
DITTO WITH FAIRED TIPS	▽	△		▷	◁
TAPERED			◇		
ELLIPTIC	⊙				



THE EFFECT OF ASPECT RATIO ON THE MAXIMUM LIFT OF FLAT PLATES OF DELTA PLANFORM.

$\Delta C_{L_{MAX}}$ = INCREMENTAL CHANGE IN $C_{L_{MAX}}$ RELATIVE TO SECTIONAL VALUE AS DATUM.

A = EFFECTIVE WING ASPECT RATIO.

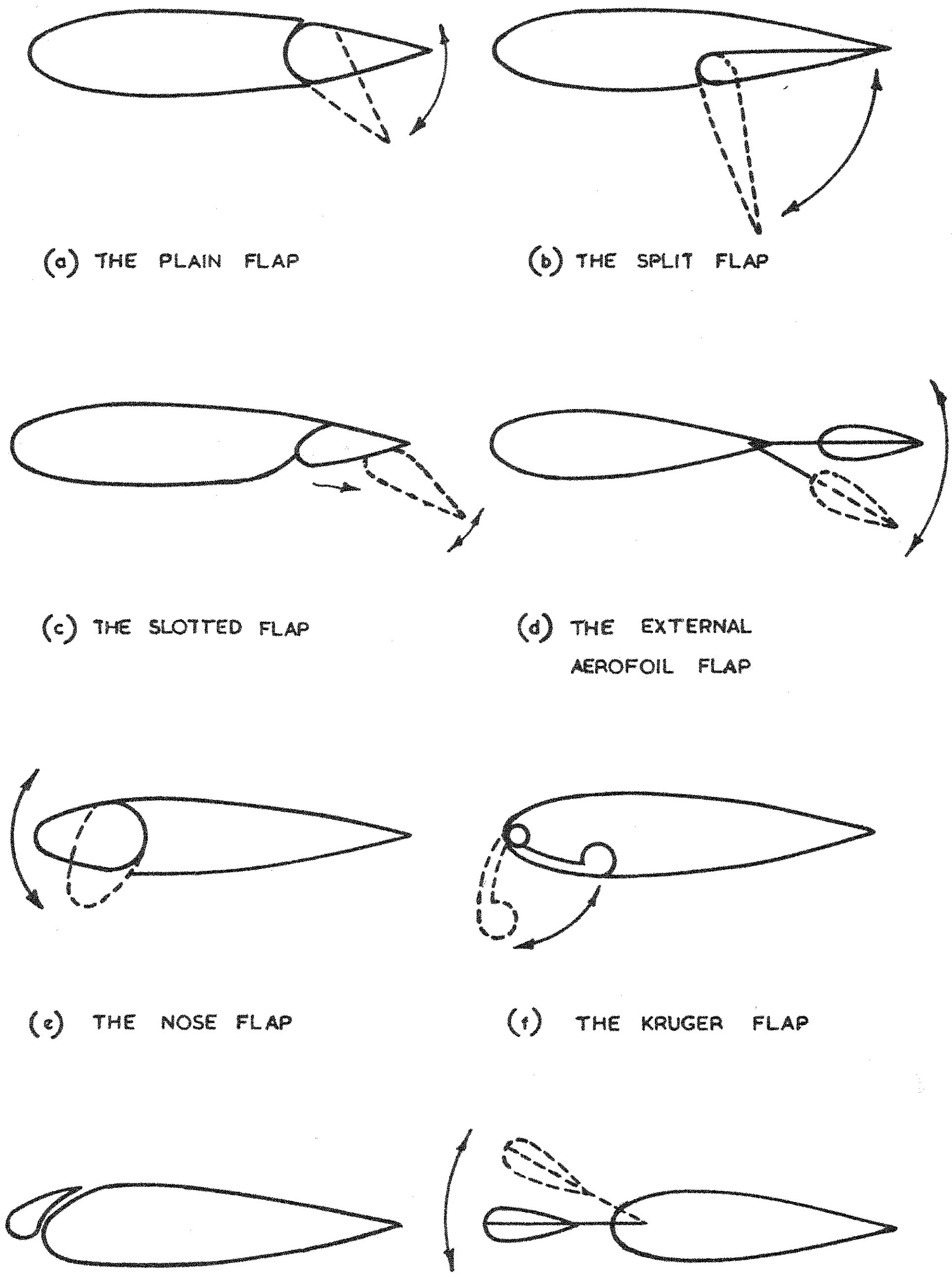
FIG. 9.

KEY TO DATA

△ DELTA. OR TRIANGULAR PLANFORM

⌂ "CROPPED" DELTA OR TRAPEZOIDAL PLANFORM (TAPER RATIO $^1/_2$)

⊙ HALF-ELLIPTICAL PLANFORM



(a) THE PLAIN FLAP

(b) THE SPLIT FLAP

(c) THE SLOTTED FLAP

(d) THE EXTERNAL AEROFOIL FLAP

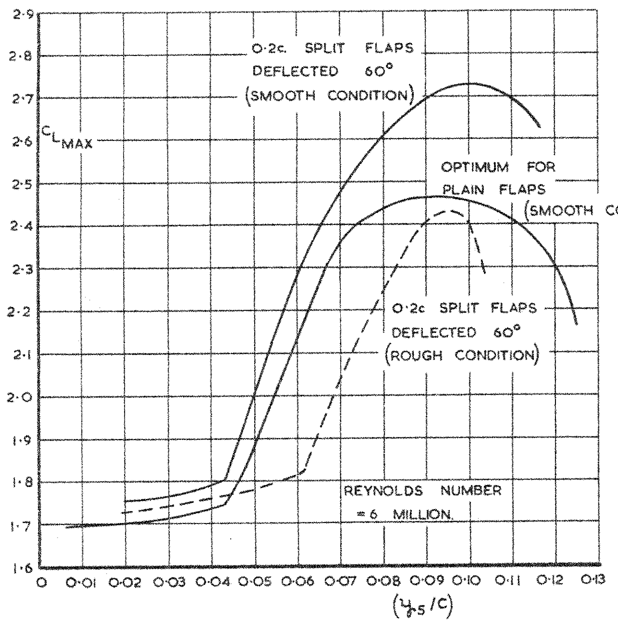
(e) THE NOSE FLAP

(f) THE KRUGER FLAP

(g) THE LEADING EDGE SLOT

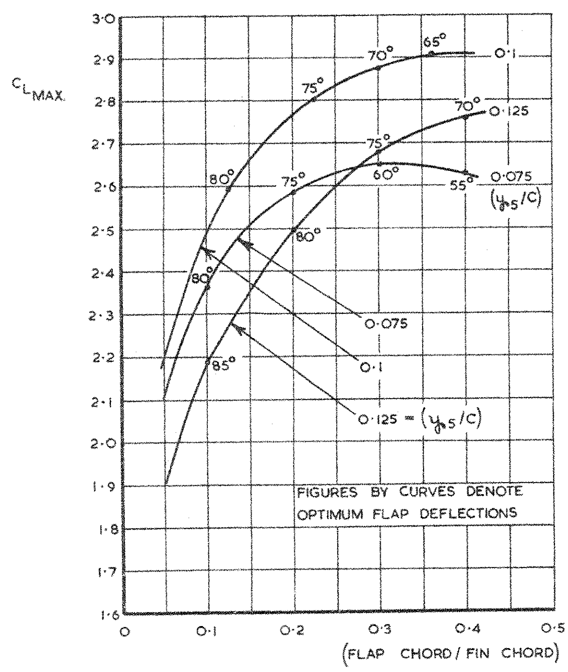
(h) THE FRONT AUXILARY AEROFOIL

DIAGRAMMATIC REPRESENTATION OF SOME TYPES OF HIGH LIFT AID



A COMPARISON OF THE SECTIONAL MAXIMUM LIFT COEFFICIENT WITH PLAIN AND SPLIT FLAPS ON SYMMETRICAL AEROFOILS
(y_5 = WING THICKNESS AT 0·05 OF THE CHORD AFT OF THE NOSE
c = WING CHORD)

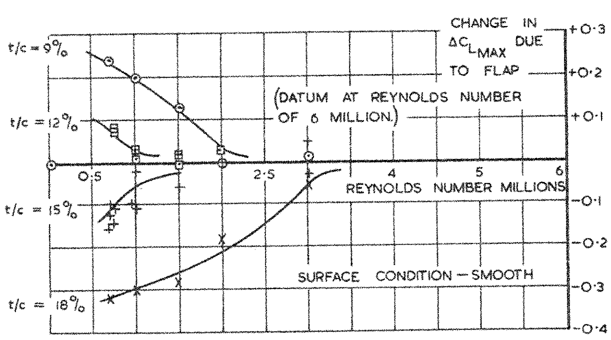
FIG. 11.



THE SECTIONAL MAXIMUM LIFT COEFFICIENT OBTAINED BY OPTIMUM DEFLECTION OF SPLIT FLAPS OF VARIOUS CHORDS

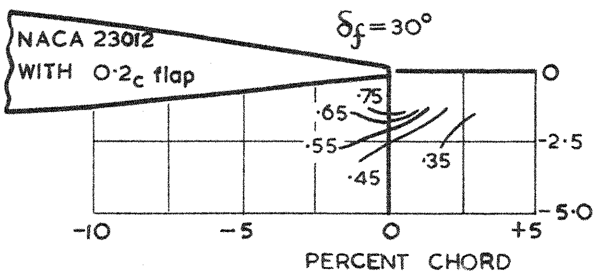
SECTION SHAPE — SYMMETRICAL
REYNOLDS NUMBER — 6 MILLION
SURFACE CONDITION — SMOOTH
(y_5 = THICKNESS OF SECTION 5% OF THE CHORD AFT OF THE NOSE
C = WING CHORD)

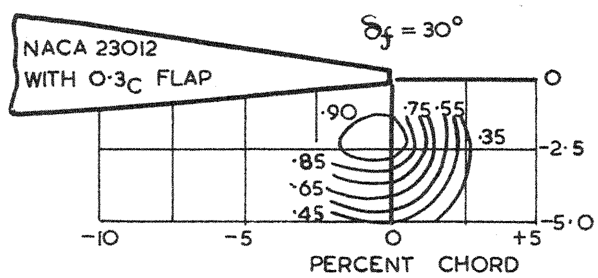
FIG. 12.

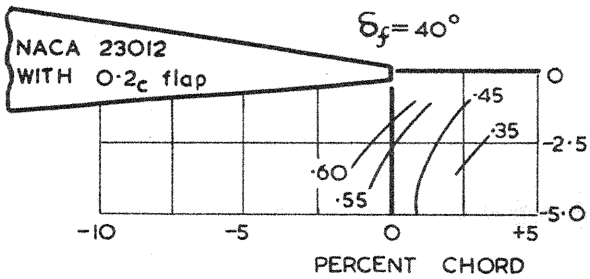


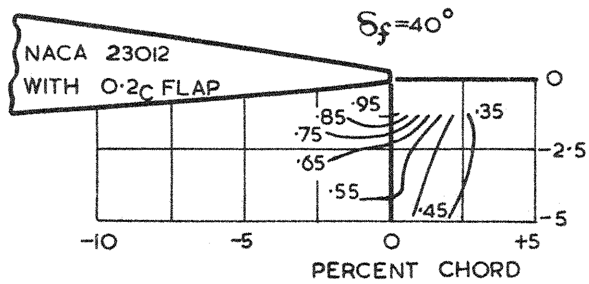
VARIATION WITH REYNOLDS NUMBER OF THE INCREMENT IN SECTIONAL MAXIMUM LIFT DUE TO A SPLIT FLAP

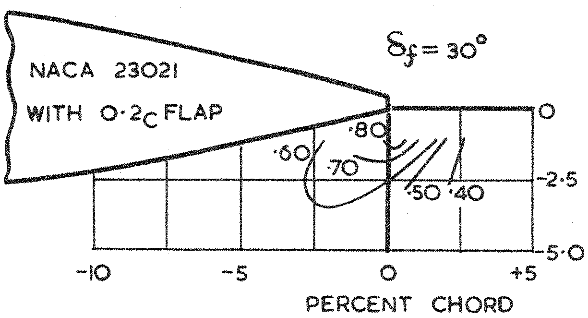
FIG. 13.









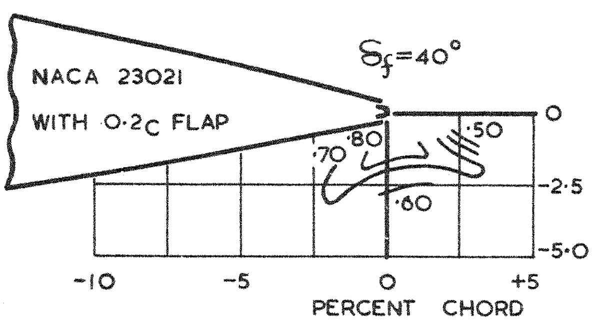


δ_f = FLAP DEFLECTION ANGLE

C = FIN CHORD (INCLUDING FLAP)

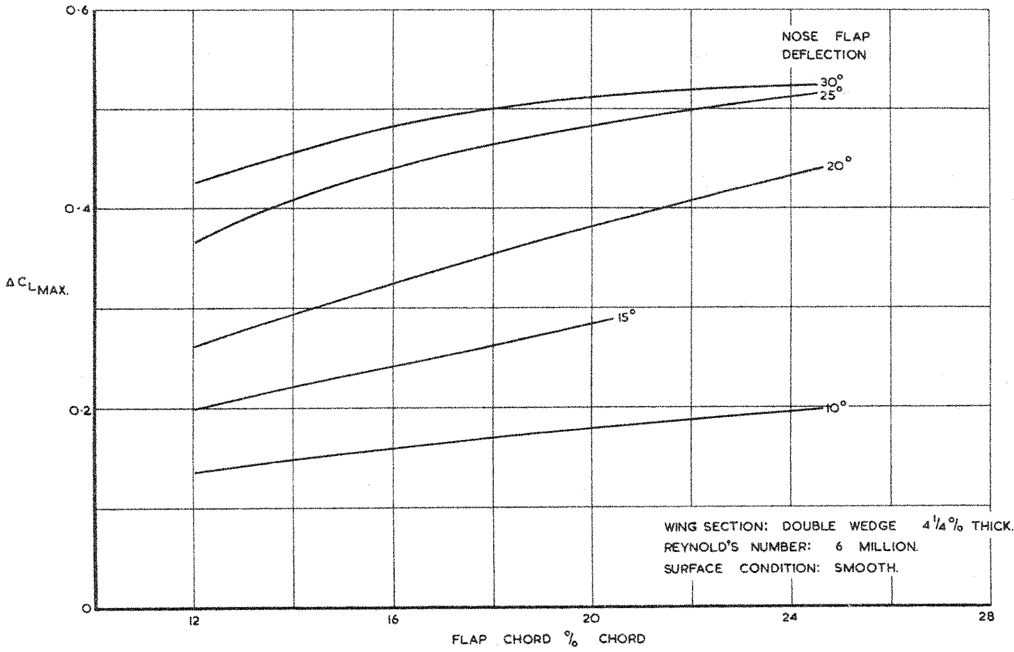
REYNOLDS NUMBER = 0·61 MILLION

SECTION OF EXTERNAL AIROFOIL = N.A.C.A. 23012.



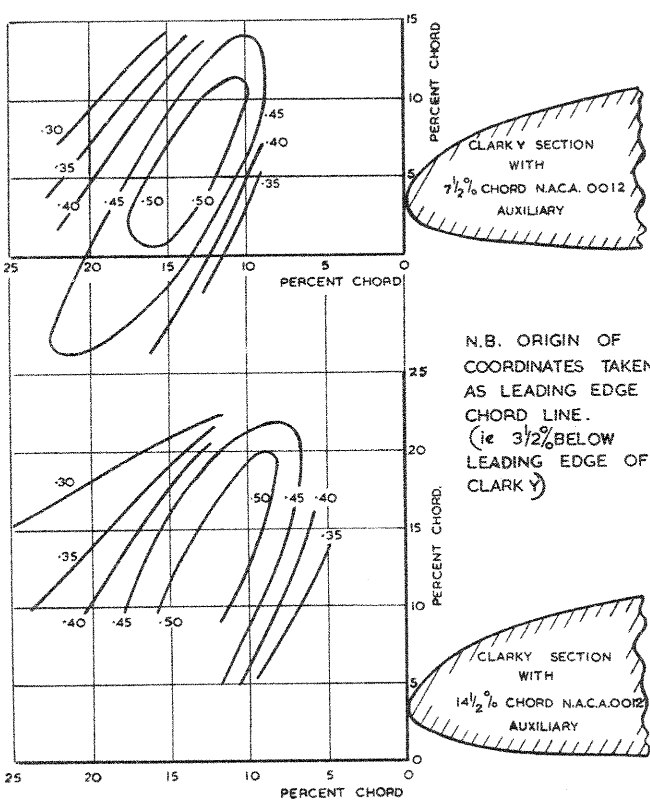
PERFORMANCE OF EXTERNAL AEROFOIL FLAPS

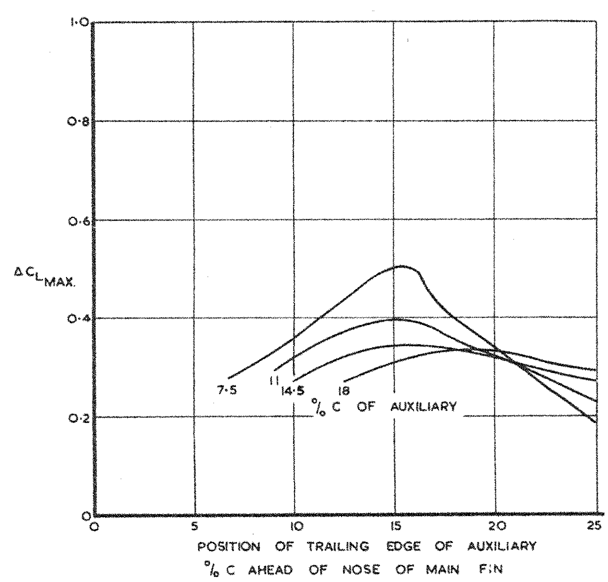
CONTOURS SHOWING INCREMENT IN SECTIONAL MAXIMUM LIFT COEFFICIENT AS INFLUENCED BY POSITION OF LEADING EDGE OF FLAP WITH RESPECT TO TRAILING EDGE OF MAIN WING



THE SECTIONAL MAXIMUM LIFT INCREMENT DUE TO DEFLECTION OF PLAIN NOSE FLAPS OF VARIOUS CHORDS

FIG. 15.





PERFORMANCE OF FRONT AUXILIARY AEROFOIL FLAPS FIG. 16.

CONTOURS SHOWING INCREMENT IN SECTIONAL MAXIMUM LIFT AS INFLUENCED BY TRAILING EDGE POSITION OF AUXILIARY AEROFOIL
N.B. $C_{L_{MAX}}$ FOR PLAIN SECTION BASED ON SECTION CHORD
$C_{L_{MAX}}$ WITH AUXILIARY BASED ON MAIN CHORD AND AUXILIARY CHORD
REYNOLDS NUMBER OF TESTS = 0·61 MILLION

INCREMENT IN SECTIONAL MAXIMUM LIFT COEFFICIENT CAUSED BY FIXED FRONT AUXILIARY AEROFOILS OF VARIOUS CHORDS AS FUNCTION OF GAP SIZE FIG. 17.

AUXILIARY PLACED AT ZERO INCIDENCE ON CHORD LINE OF MAIN WING.
$\Delta C_{L_{MAX}}$ = INCREMENT IN $C_{L_{MAX}}$ CAUSED BY ADDITION OF AUXILIARY. $C_{L_{MAX}}$ FOR PLAIN SECTION BASED ON SECTION CHORD.
$C_{L_{MAX}}$ WITH AUXILIARY BASED ON MAIN CHORD & AUXILIARY CHORD